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RESEARCH MEMORANDUM

EXPERIMENTAL PERFORMANCE OF CHLORINE TRIFLUORIDE - HYDRAZINE PROPELLANT COMBINATION IN 100-POUND-THRUST ROCKET ENGINE

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS
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SUMMARY

The experimental performance of chlorine trifluoride and hydrazine was measured over a range of propellant mixtures in a 100-pound-thrust rocket engine. The engine was operated at a combustion-chamber pressure of approximately 300 pounds per square inch absolute. Experimental values of specific impulse, volume specific impulse, heat rejection per propellant weight, nozzle thrust coefficient, and characteristic velocity were obtained as functions of percent fuel by weight in the propellant mixture.

The maximum measured specific impulse of 234 pound-seconds per pound occurred at a mixture of 33-percent fuel by weight. When corrected for heat rejection to the engine walls, the maximum specific impulse was 247 pound-seconds per pound, 98-percent of the estimated theoretical value for the fuel and the nozzle used. The measured volume specific impulse reached a maximum of 331 $\times$ 62.4 pound-seconds per cubic foot and when corrected for the heat transfer increased to 349 $\times$ 62.4 pound-seconds per cubic foot, both at a mixture ratio containing 33-percent fuel by weight.

The values of heat rejection attained a maximum of approximately 245 Btu per pound of propellant mixture, which corresponds to an over-all heat-flux density of 1.7 Btu per second per square inch, in the mixture range of 30- to 38-percent fuel by weight. The average nozzle thrust coefficient was 1.34 and the maximum characteristic velocity was 5590 feet per second.

The trends, as indicated by the experimental-performance and the heat-rejection curves, were markedly different from what was predicted by theoretical calculations.

INTRODUCTION

The use of hydrazine and chlorine trifluoride as a rocket-propellant combination is of interest because of the high thrust per propellant volume, the mutual self-ignitibility of the propellants, the high boiling point, and the stabilities that permit the propellants to be stored for reasonable lengths of time.

The performance of chlorine trifluoride and hydrazine has been calculated by Douglas Aircraft Co., Inc., Jet Propulsion Laboratory of the California Institute of Technology, Battelle Memorial Institute, and Bell Aircraft Corp. A summary of most of these calculations is presented in reference 1.

Chlorine trifluoride as an oxidizer for hydrazine gives higher performance than other oxidizers capable of indefinite storage, such as hydrogen peroxide and nitric acid. The calculated specific-impulse and volume-specific-impulse values (references 2 and 3) of these and other oxidizers with hydrazine are shown in the following table:

Oxidizer	Boiling point (°F)	Maximum specific impulse (lb-sec/lb)	Volume specific impulse (lb-sec/cu ft)
Chlorine trifluoride, ClF_3	52	263	391
Nitric acid, HNO_3	187	245	310
Hydrogen peroxide, H_2O_2	306	249	305
Oxygen, O_2	-297	270	290
Ozone, O_3	-170	283	366
Fluorine, F_2	-305	310	328

The last three oxidizers give higher specific impulse than chlorine trifluoride but lower volume specific impulse. Among other disadvantages, they also have very low boiling points. Other factors, such as engine application, heat transfer, handling, and availability, must be considered in comparing propellant combinations but the advantages mentioned for chlorine trifluoride and hydrazine make an experimental evaluation of the performance of this combination desirable.

As part of the general NACA program on rocket propellants, the experimental performance of a 100-pound-thrust rocket engine using chlorine trifluoride and hydrazine was determined over a range of propellant mixtures at a combustion-chamber pressure of 300 pounds per square inch absolute. The results of experiments conducted at the NACA Lewis laboratory from August to December 1948 are presented herein.

APPARATUS

A diagrammatic sketch of the apparatus and the flow systems is shown in figure 1. Charcoal-purified helium under pressure, controlled by two-stage regulation, was used to force the propellants into the combustion chamber. Each tank was suspended from a counterbalanced weighing beam, which was used for the measurement of propellant flow. The rocket engine was mounted on a thrust stand. The engine was inclined downward at an angle of 30° to horizontal in order to prevent the accumulation of the propellants in the combustion chamber. A photograph of part of the apparatus is shown in figure 2.

Engine Assembly

A cross-sectional view of the engine assembly is shown in figure 3. The engine assembly consisted of an injection plate and a water-cooled combustion chamber and exhaust nozzle. The monel injection plate was fitted with four removable injector nozzles inserted into the inside face. The two hydrazine injectors and two chlorine trifluoride injectors produced solid jets, which impinged at a common point 1.1 inch from the outlets of the injector nozzles. The injector orifices were cylindrical and for most runs had beveled inlet sections. The inside diameters of the fuel nozzles were either 0.041 or 0.051 inch; the oxidant nozzles were 0.061 or 0.070 inch, depending upon the mixture that was desired. A pressure tap was located in the injection plate to obtain combustion pressure.

The combustion-chamber and nozzle walls were 1/8-inch-thick nickel. The engine had a length-to-diameter ratio of 3.4, a ratio of combustion-chamber diameter to throat diameter of 4.1, and a characteristic length (ratio of combustion-chamber volume to exhaust-nozzle throat area) L^* of 137 inches. The rather large characteristic length L^* was chosen to insure complete combustion. The nozzle was designed to obtain optimum performance for an expansion ratio of 20.4 with a specific heat ratio of 1.3.

Instrumentation

The oxidant and fuel tanks were suspended from standard beam balances and each was equipped with a counterbalance tank. The 1140
counterbalance tanks were containers that could be remotely filled with water or emptied as required to balance the propellant tank initially. A photograph of one of the weighing systems is presented in figure 4. Unbalanced forces caused by changes of weight in the propellant tank during a run were transmitted to a cantilever spring that was designed for a total deflection force of 4 pounds. The strain gages cemented on the cantilever spring were connected by a resistance-bridge circuit to a continuous-recording self-balancing potentiometer. The accuracy of the weight measurement was about 1 percent. Two or more calibrations of each weighing system were made before each run and these calibrations, which checked to within 1 percent, were used for that run. In two instances these calibrations were repeated immediately after the run and in both instances the same results were obtained. During the complete investigation, the dead-weight calibration constants (lb/space) for the oxidant system were found to vary less than 3 percent, whereas for the fuel system the dead-weight calibrations varied less than 1.5 percent.

The thrust produced by the engine was measured by means of a cantilever spring equipped with strain gages. The strain gages were connected in a resistance-bridge circuit to a rapidly self-balancing recording potentiometer. The accuracy of the thrust measurement was 1 percent. Dead-weight calibrations were made before each run and the calibration used for the run. In two instances the calibrations were repeated immediately after the run and in both instances the same results were obtained. Over the entire period of experimentation, the mean variation of the thrust calibration constant (lb/space) was less than 1 percent.

Combustion-chamber pressure was measured by a Bourdon-tube pressure recorder. The accuracy of the recorder for static-pressure conditions was higher than 1.5 percent.

The coolant-water flow was measured by an adjustable orifice equipped with an electric transmitter and recorder. The accuracy of the system was higher than 2 percent. Iron-constantan thermocouples located in the inlet and the outlet of the engine coolant jacket were used to measure the coolant-water temperatures. The temperatures were recorded on self-balancing continuous-recording potentiometers. The full-scale accuracy was 1.0 percent.

In addition to the errors of measurement of weight for determining flow rates, an additional error is involved in the interpretation of the weight records. Because the flow rates are obtained from the time-weight records, small fluctuations in the record can cause an appreciable error in the determination of the slope. The maximum error observed in the determination of slope was 4 percent. From analysis of the data records, however, the probable error in slope determination for all the runs is approximately 1 percent. The probable error in the specific-impulse values, which combine the measurement and the interpretation errors of thrust and propellant flow rates, is about 3 percent.

PROPELLANTS

The oxidant used was commercial chlorine trifluoride stated by the manufacturer to be at least 99.5 percent pure. The fuel was 96-percent hydrazine and 4-percent water. The purity of the hydrazine was determined by the method described in reference 4.

SYMBOLS

The following symbols were used in the calculations and on the figures:

A_t rocket exhaust-nozzle-throat cross-sectional area, (sq. in.)

C_F nozzle thrust coefficient, dimensionless, $(F/P_c A_t)$

c^* characteristic velocity, (ft/sec), $(P_c A_t g/w)$

F thrust, (lb)

g gravitational constant, 32.17 (ft/sec^2)

I specific impulse, (lb-sec/lb)

I_d volume specific impulse, (lb-sec/cu ft) $\times$ 62.4

J mechanical equivalent of heat, 778 (ft-lb/Btu)

L^* characteristic length (combustion-chamber volume/throat area), (in.)

P_c combustion-chamber pressure, (lb/sq in. absolute)

Q heat rejection per propellant weight, (Btu/lb)

T_c combustion-chamber temperature, (^{o}R)

w total propellant flow, (lb/sec)

η ideal thermal cycle efficiency, dimensionless 1140

λ nozzle-angle correction factor, dimensionless

PROCEDURE

The rocket engine was started by simultaneously injecting the chlorine trifluoride and the hydrazine; ignition resulted immediately. The average duration of a run was 8 seconds. Runs were made with various propellant mixtures and at combustion pressures of approximately 300 pounds per square inch absolute. Immediately after the run, the propellant systems were purged with helium.

With the comparatively large characteristic-length L^* engine used, an appreciable amount of heat was lost to the engine walls. The heat absorbed was determined by the product of the temperature rise of the coolant water, the specific heat of the water, and the water flow. In order to estimate the experimental performance of the propellants without heat loss, the measured specific-impulse values were corrected for heat loss. The approximate correction was based on the assumption that the heat loss, if available, could be converted into kinetic energy at the ideal cycle efficiency. Thus

$$I\ (\text{corrected}) = \sqrt{I^2\ (\text{experimental}) + \frac{2J}{g} Q\eta}$$

A value of 0.51 was used for η in all the corrections.

A comparison of the experimental performance with theoretically calculated values is desirable. Inasmuch as such calculations as those presented in reference 1 are generally for ideal conditions that include pure propellants and parallel flow of exhaust gases in nozzles, the theoretical data were adjusted to represent theoretical performance for the fixed experimental conditions. The hydrazine used contained 4-percent water and the nozzle had a divergent half-angle of 15°. The theoretical specific impulse for the fuel as used was estimated as 2 percent lower than the performance of pure anhydrous hydrazine with chlorine trifluoride, whereas corrections for the divergence from axial flow through the exhaust nozzle lowered the theoretical performance another 2 percent (reference 5).

The values of thrust coefficient C_F were determined from the experimental data, that is,

$$C_F = \frac{F}{P_c A_t}$$

Characteristic velocity c^* was also calculated from experimental data, that is,

$$c^* = \frac{P_c A_t g}{w}$$

Experimental volume-specific-impulse values as measured and as corrected for heat rejection were obtained from the equivalent specific-impulse values and the average bulk density of the propellants at the corresponding mixtures. The average densities were calculated from data in reference 3.

RESULTS AND DISCUSSION

Theoretical and experimental specific-impulse values for chlorine trifluoride and hydrazine as functions of percent fuel by weight in the propellant mixture are shown in figure 5. The experimental specific impulse covers a propellant mixture range from approximately 20- to 55-percent fuel by weight. The data show a rapid change in specific impulse in the mixture range of 30- to 38-percent fuel by weight with a maximum of 234 pound-seconds per pound at a fuel-rich mixture of 33-percent fuel by weight (stoichiometric, 25.7-percent fuel by weight for pure propellants). Most of the data were obtained at a combustion pressure of 300 pounds per square inch absolute; several runs made at lower and higher pressures (275 and 340 lb/ sq in.) show no appreciable change in specific impulse for the pressure range covered.

Corrections to the experimental specific impulse for heat loss accentuated the rapid change of specific impulse in the mixture range of 30- to 38-percent fuel by weight, and gave a maximum value of 247 pound-seconds per pound at a mixture of 33-percent fuel by weight.

For comparison with experimental data, theoretical specific-impulse data for equilibrium expansion are also shown in figure 5.

The first curve is from reference 2 and the second theoretical curve represents the same data adjusted to correspond more nearly to the fixed conditions of the experiments (96-percent hydrazine and 15° half-angle of divergent nozzle). A comparison of the theoretical data estimated for the fuel and the nozzle used and the experimental data corrected for heat rejection shows that experimental data gave high performance in the fuel-rich region, 32- to 35-percent fuel (approximately 98 percent of the theoretical value at 33-percent fuel by weight). In the stoichiometric region, the experimental data were much lower than theoretically predicted (76 percent of the theoretical value at stoichiometric).

The reasons for these differences between theoretical and experimental results are unknown and a satisfactory explanation would require further investigation. The differences may be caused by the particular injector system used for the experiments. The disagreement noted among the theoretical data (reference 1) would not account for the experimental differences just mentioned.

One of the principal advantages of the chlorine trifluoride - hydrazine rocket-propellant combination is the relatively high densities of the propellant mixtures. The effect of these high densities are reflected by the volume-specific-impulse data shown in figure 6, in which theoretical and experimental values are shown as functions of propellant mixture. The data are treated in the same manner as the specific-impulse data of figure 5. Again a narrow region of maximum experimental performance is shown between 30- and 38-percent fuel by weight. The maximum measured and corrected experimental volume-specific-impulse values were 331 and 349 × 62.4 pound-seconds per cubic foot, respectively, at 33-percent fuel by weight. The comparison between theoretical and experimental values of volume specific impulse are the same as for the specific impulse inasmuch as at a given propellant mixture the specific impulse and volume specific impulse differ only by a constant.

Theoretical combustion-chamber temperature and experimental heat rejection as functions of mixture ratio are shown in figure 7. The heat-rejection data are constant at about 138 Btu per pound in the region near stoichiometric from 20- to 29-percent fuel. At 29- to 30-percent fuel, however, the heat rejection increased sharply and from 30- to 38-percent fuel (the region of maximum performance) it averaged about 240 Btu per pound of propellant. At about 38-percent fuel, the heat rejection fell sharply; the data in the fuel-rich region from 38 to 46 percent were scattered. The maximum heat rejection of 245 Btu per pound corresponds to an over-all heat-flux density of 1.7 Btu per second per square inch.

The sudden increase and drop in the heat-rejection data would not be expected from the curve of theoretical combustion-chamber temperature, which shows a maximum of 6800° R at approximately 26-percent fuel, and about 15-percent decrease between 26- and 38-percent fuel by weight. The rapid variation of heat rejection follows closely the rapid variation of specific impulse (fig. 5) and further indicates the marked change in performance of the engine in the mixture region of 30- to 38-percent fuel by weight.

Measured values of nozzle thrust coefficient C_F and characteristic velocity c^* plotted as functions of percent fuel by weight are shown in figure 8. The C_F data for the combustion-chamber pressure range of 300 ± 15 pounds per square inch absolute show considerable scatter. The average was 1.34 with a mean deviation of 2.5 percent. A theoretical value of 1.36 was calculated for C_F using a specific heat ratio of 1.31, an expansion ratio of 20.4, a nozzle-area ratio of 3.19, and a nozzle correction factor λ of 0.98. For the runs made at a combustion-chamber pressure of 300 ± 15 pounds per square inch absolute, a maximum c^* of about 5590 feet per second occurred at a mixture of about 36-percent fuel by weight. As with the C_F values, considerable scatter in the data occurred. The values of c^* reflect, in general, the performance trends shown by the specific-impulse data in figure 5; that is, maximum performance in the fuel-rich-mixture region and low performance in the stoichiometric region. No close correlation was obtained between specific impulse and characteristic velocity, because of the scatter in nozzle thrust coefficient.

Throughout the investigation, no serious difficulties were encountered in handling the propellants or operating the equipment. The propellants ignited spontaneously and the combustion appeared to be smooth. Examination of color motion pictures taken during the runs showed the flame to be only slightly luminous, with a series of shock waves throughout the entire length of approximately $2\frac{1}{2}$ feet. In some cases white smoke appeared about $3\frac{1}{2}$ feet from the end of the flame; this smoke was to be expected because of the highly hygroscopic properties of some of the exhaust products. After some of the runs, especially in the fuel-rich region, a light deposit was found on the inner walls of the engine. Qualitative analysis of one sample of deposit indicated that it may have been mostly hydrazine hydrochloride.

SUMMARY OF RESULTS

The following results were obtained from measuring the experimental performance of chlorine trifluoride and hydrazine in a 100-pound-thrust rocket engine over a range of propellant mixtures from 20- to 55-percent fuel by weight. The engine had a characteristic length (ratio of combustion-chamber volume to exhaust-nozzle-throat area) of 137 and was operated at a combustion-chamber pressure of approximately 300 pounds per square inch absolute.

1. The maximum measured specific impulse was 234 pound-seconds per pound and occurred at a mixture of 33-percent fuel by weight (stoichiometric, 25.7 percent fuel by weight). When corrected for heat losses the maximum specific impulse was 247 pound-seconds per pound. The corrected value is 98 percent of the estimated theoretical value for the fuel and the nozzle used.

2. The trend of the experimental curve differs considerably from the theoretical curve. The highest experimental performance occurred in the mixture region of 30- to 38-percent fuel by weight. When compared with theoretical values, the experimental performance was low in the stoichiometric region.

3. The measured volume specific impulse reached a maximum of 331×62.4 pound-seconds per cubic foot at a mixture of 33-percent fuel by weight. When corrected for heat losses the maximum volume specific impulse was 349×62.4 pound-seconds per cubic foot.

4. The curve of heat rejection as a function of propellant mixture followed the same trends as the experimental specific-impulse curve. The maximum heat rejection, in the mixture range of 30- to 38-percent fuel by weight, was approximately 245 Btu per pound of propellant mixture, which corresponds to an over-all heat-flux density of 1.7 Btu per second per square inch.

5. The average nozzle thrust coefficient was 1.34 and the maximum characteristic velocity was 5590 feet per second at a mixture of 36-percent fuel by weight.

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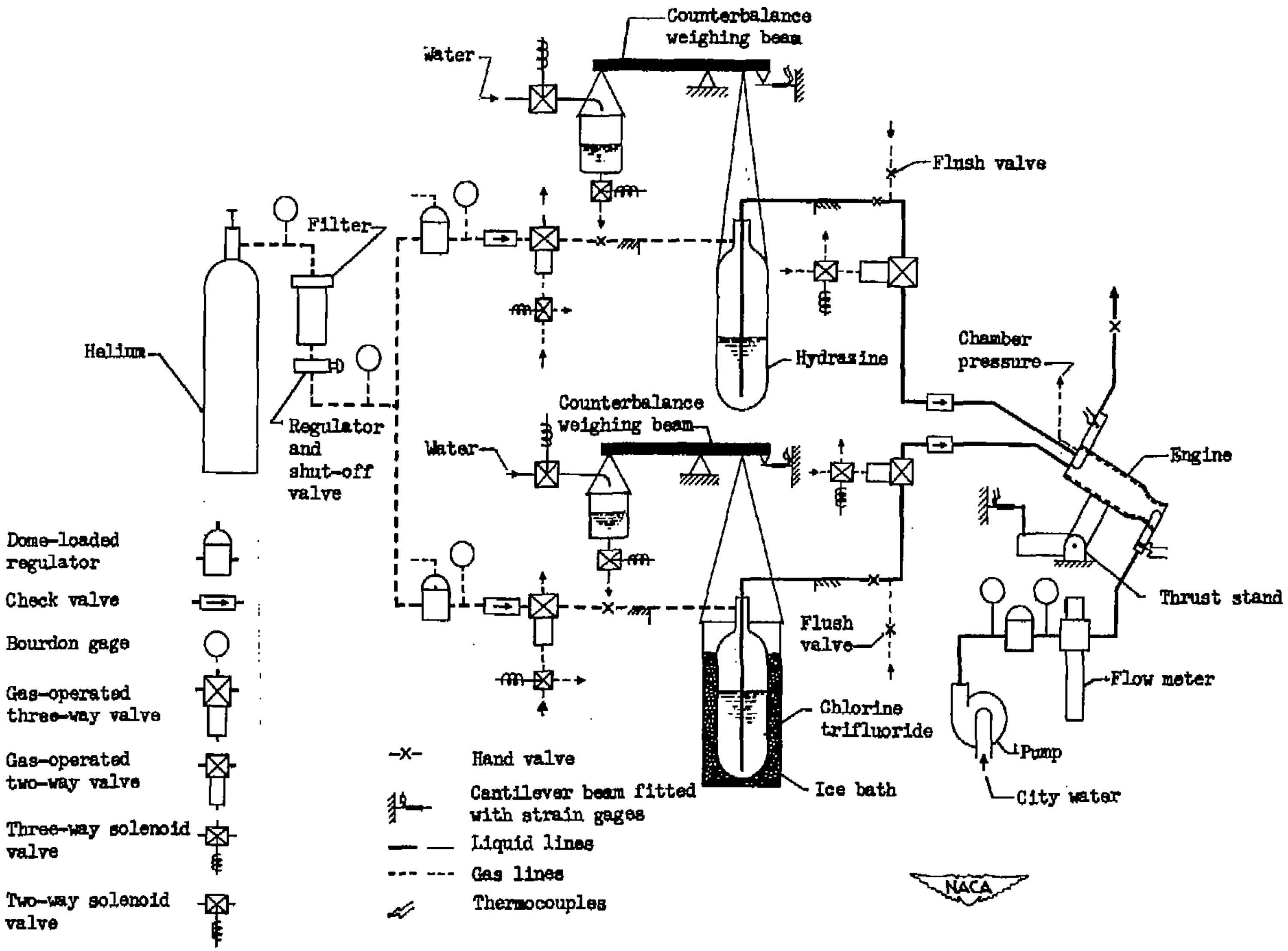

Figure 1. - Diagrammatic sketch of 100-pound-thrust rocket apparatus for investigation of chlorine trifluoride and hydrazine.

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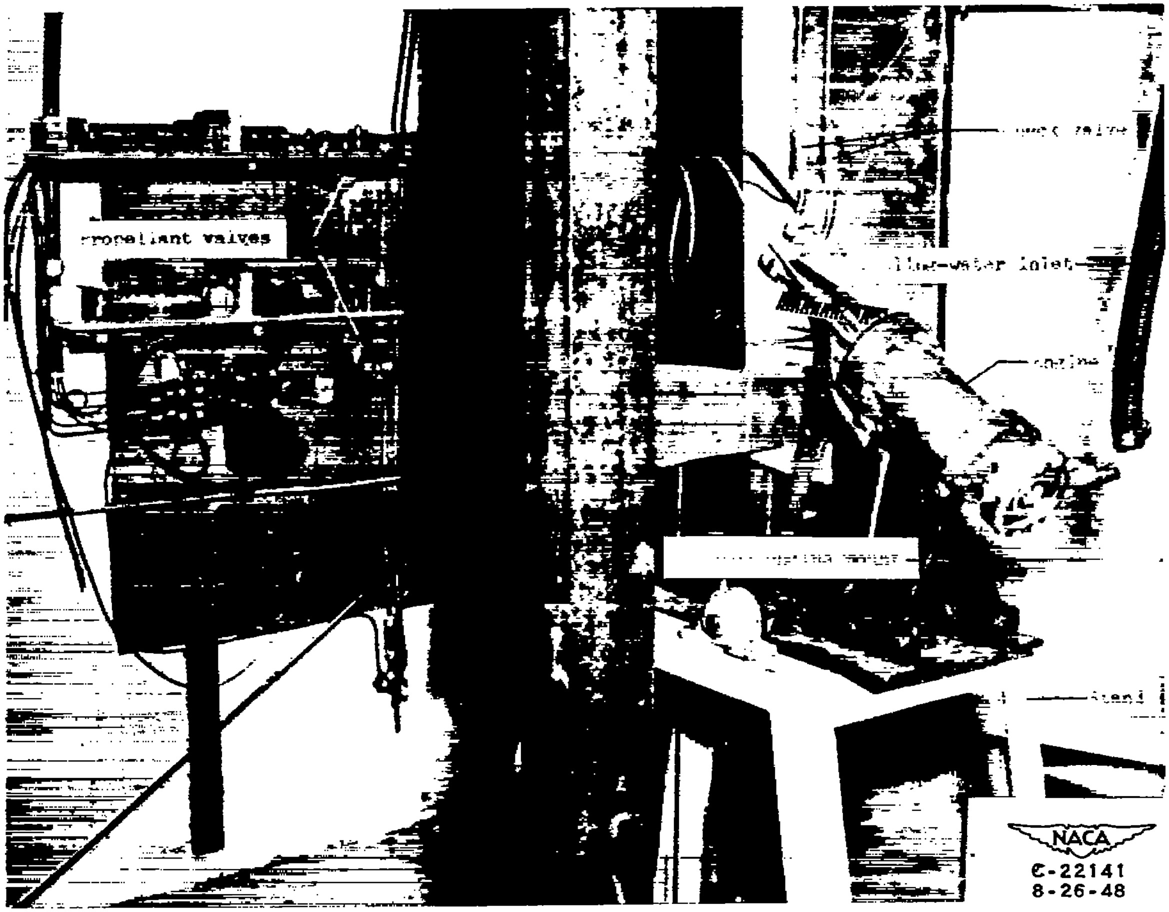

Figure 2. - Rocket engine mounted on thrust stand.

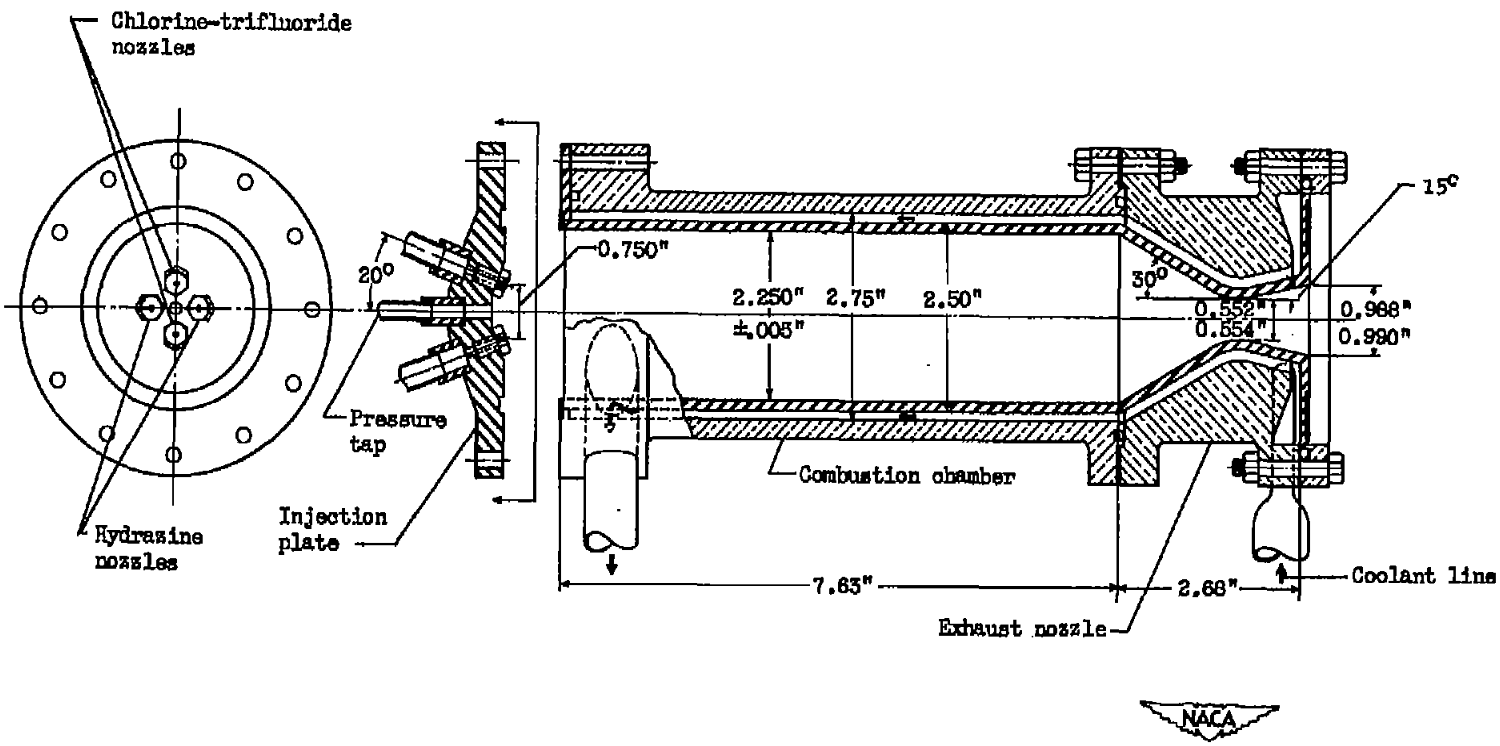

Figure 3. - Rocket-engine assembly.

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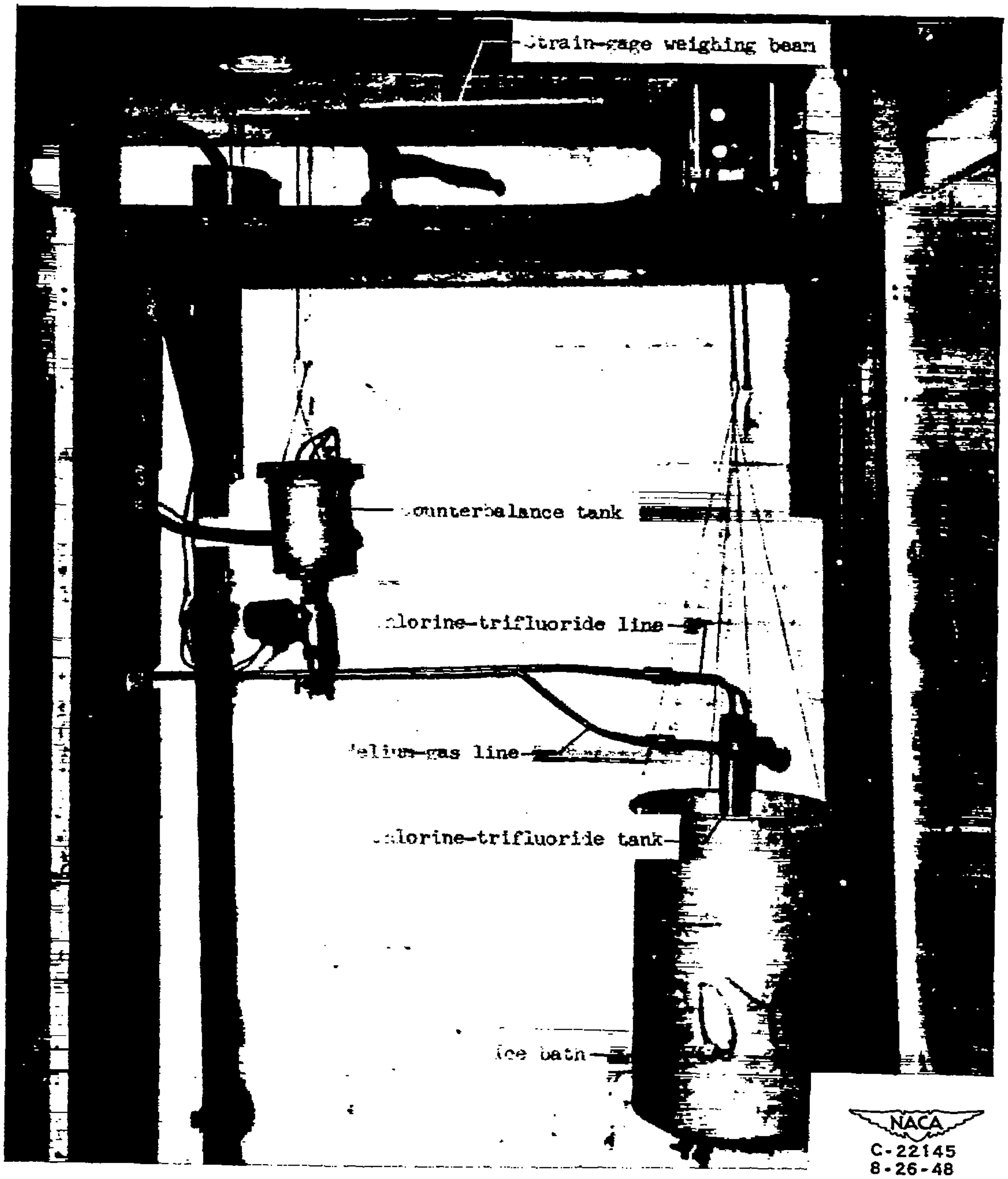

Figure 4. - Chlorine trifluoride propellant weighing system.

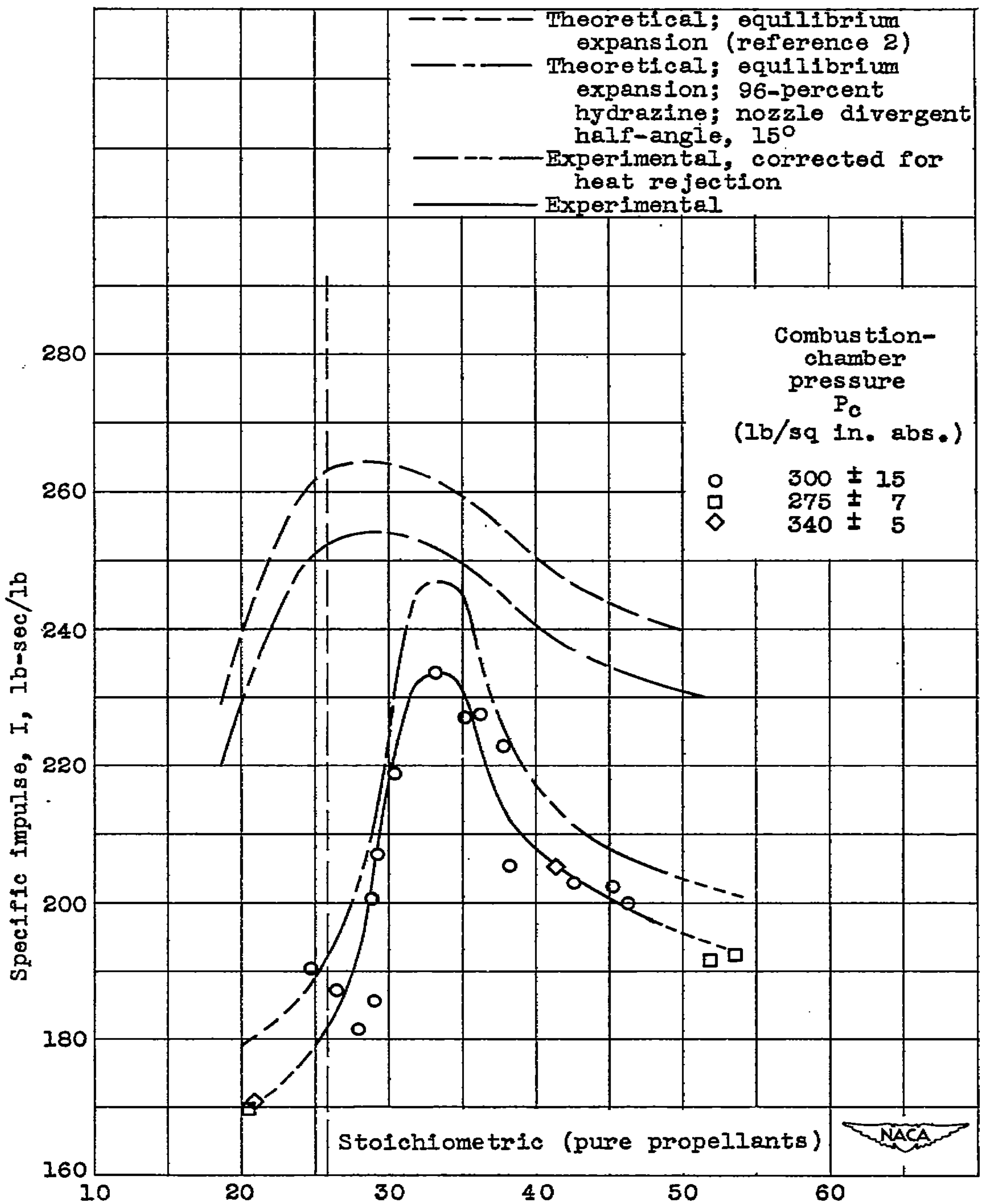

Figure 5. - Theoretical and experimental specific impulse of 100-pound-thrust rocket engine using chlorine trifluoride and hydrazine.

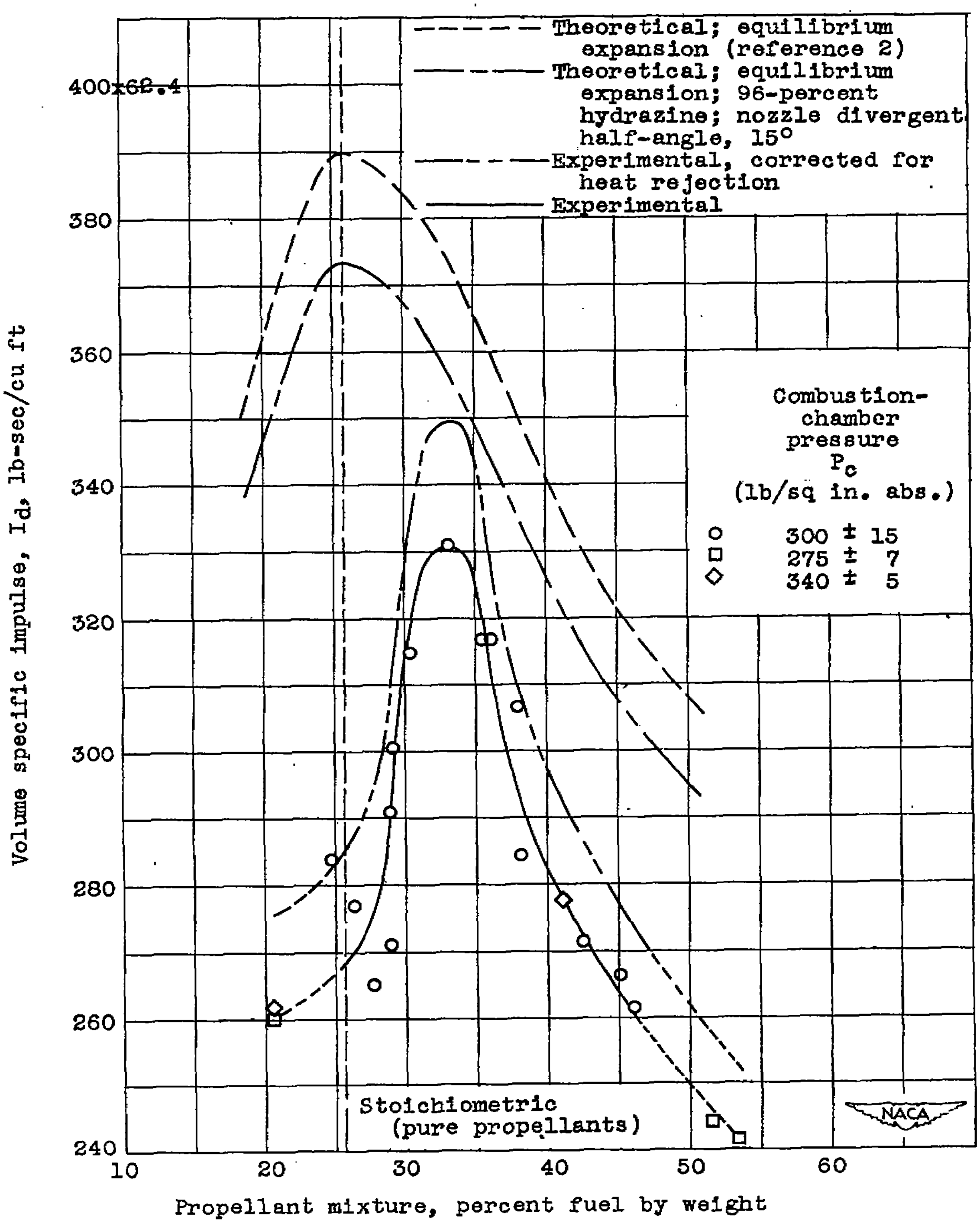

Figure 6. - Theoretical and experimental volume specific impulse of 100-pound-thrust rocket engine using chlorine trifluoride and hydrazine.

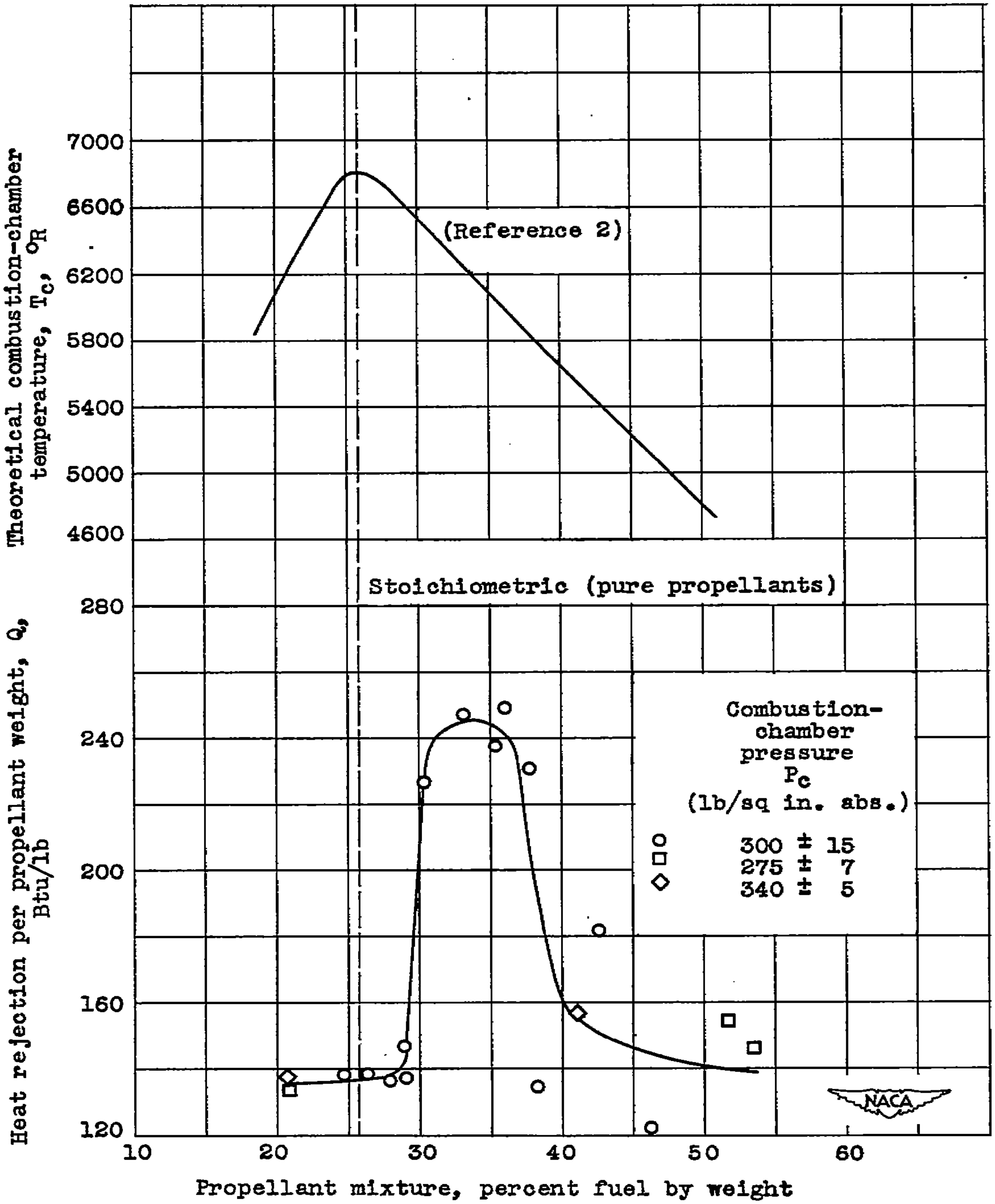

Figure 7. - Theoretical combustion-chamber temperature and experimental heat rejection in 100-pound-thrust rocket engine using chlorine trifluoride and hydrazine.

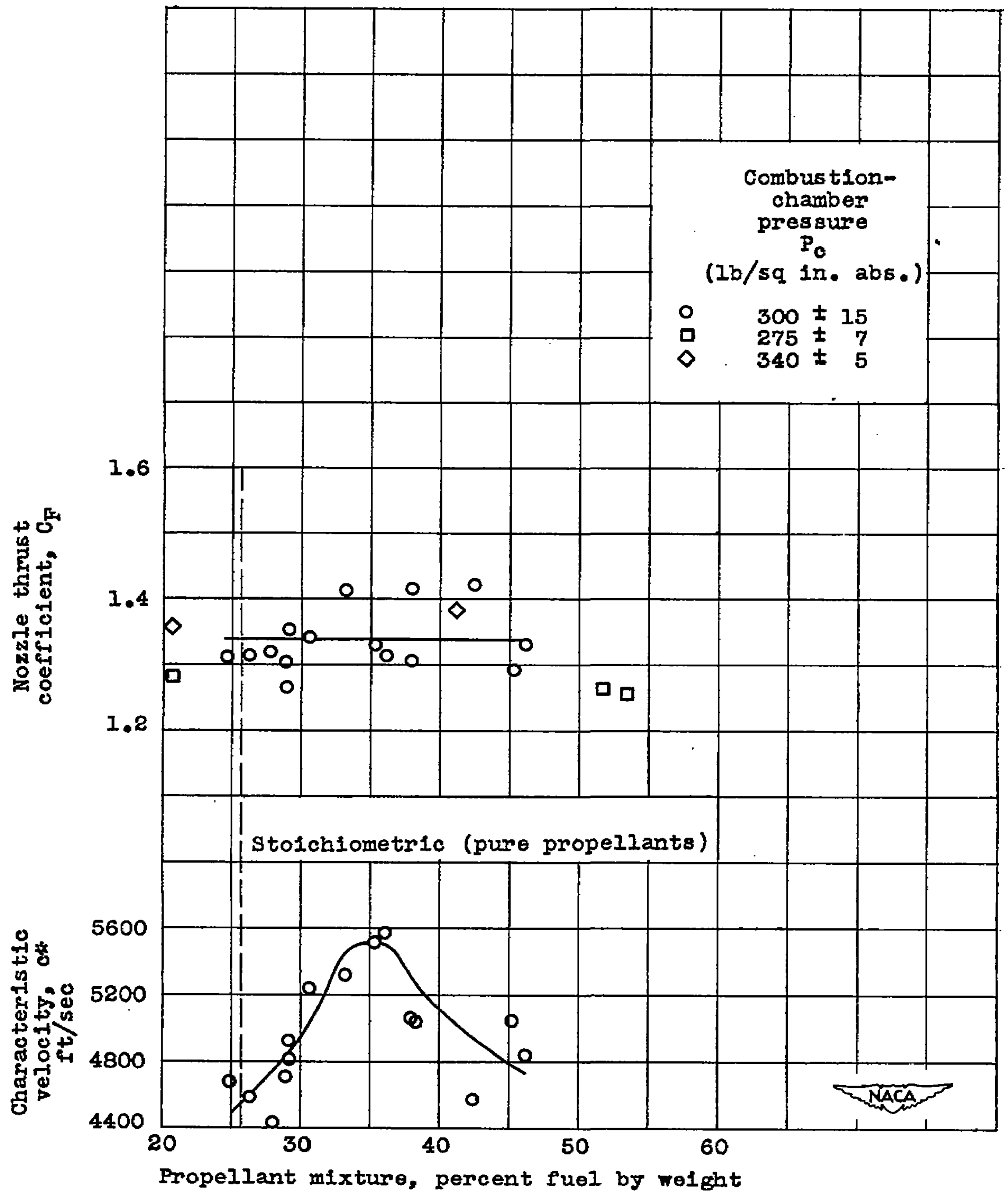

Figure 8. - Experimental nozzle thrust coefficients and characteristic velocities obtained with 100-pound-thrust rocket engine using chlorine trifluoride and hydrazine.

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